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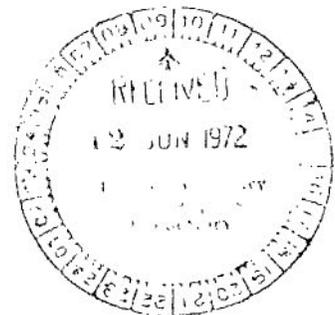
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APOLLO EXPERIENCE REPORT -
DEVELOPMENT OF GUIDANCE TARGETING
TECHNIQUES FOR THE COMMAND MODULE
AND LAUNCH VEHICLE

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16. Abstract The development of the guidance targeting techniques for the Apollo command module and launch vehicle is discussed for four types of maneuvers: (1) translunar injection, (2) translunar mid-course, (3) lunar orbit insertion, and (4) return to earth. The development of real-time targeting programs for these maneuvers and the targeting procedures represented are discussed. This document is intended to convey historically the development of the targeting techniques required to meet the defined target objectives and to illustrate the solutions to problems encountered during that development.				13. Type of Report and Period Covered Technical Note	
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ACRONYMS

AST	Abort-Scan Table
ATDLP	Apollo Trajectory Decision Logic Prototype
ATP	alternate target point
BAP	best adaptive path
CDH	constant delta height
CMC	command module computer
CSM	command and service module
EFCUA	extreme fuel-critical, unspecified area
FCUA	fuel-critical, unspecified area
IGM	Iterative Guidance Mode
LOI	lunar-orbit insertion
LOPC	lunar-orbit plane change
MIT	Massachusetts Institute of Technology
MPT	Maneuver Planning Table
MSC	Manned Spacecraft Center
MSFC	Marshall Space Flight Center
NFR	nonfree return
PTP	primary target point
RTCC	real-time computer complex
RTE	return to earth
RTED	Return-to-Earth Digital
S-IV	Saturn IV
S-IVB	Saturn IVB
TCUA	time-critical, unspecified area



TEI	transearth injection
TEMC	transearth midcourse
TLI	translunar injection
TLMC	translunar midcourse
TO	Trade-off

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SUMMARY

To develop guidance targeting techniques for the Apollo command module and launch vehicle, it was necessary to consider the onboard-guidance capability; the mission objectives, mission rules, and hardware constraints; the vehicle performance capabilities, trajectory characteristics, and crew time line; the current optimization and targeting techniques; the effect each maneuver would have upon subsequent maneuvers; and the vulnerability of each maneuver to varying initial conditions. However, in accordance with the Apollo Program time line, these concepts and techniques were being developed in parallel with the development of targeting techniques. Consequently, the targeting development had to proceed in distinct phases, and each phase was dependent upon advances made in the previous phase.

The development of the targeting procedures and associated real-time computer programs for four types of maneuvers is discussed. Each targeting program, with the exception of the Translunar-Injection Targeting-Update Program, has been used successfully to target maneuvers for each Apollo lunar mission. Although capable of fulfilling the program objectives, the Translunar-Injection Targeting-Update Program has not been needed during any Apollo lunar mission through 1969.

INTRODUCTION

Targeting Process

A series of 22 joint meetings among NASA Manned Spacecraft Center (MSC) and prime contractor personnel was held in 1965 and 1966 to investigate the operational targeting procedures for the Apollo lunar missions. During these meetings, the basic concepts and framework were formed for the development of guidance targeting techniques.

The following definitions, which were derived during these meetings, are given to clarify the terminology used in this report.

1. Targeting is the process followed to obtain guidance parameters for specific maneuvers throughout a mission.
2. Target objectives are the exact requirements for a particular mission phase, which may be a single maneuver (e. g., translunar injection (TLI)) or a series of maneuvers (e. g., translunar midcourse (TLMC) maneuvers).
3. Target parameters approximate the target objectives within an acceptable degree of accuracy.
4. Guidance parameters represent the target parameters in the guidance mechanization.

As an example, this terminology is considered in context with the TLI targeting procedure.

1. Target objective — A translunar trajectory that has a specified altitude and moon-referenced latitude at the closest approach to the moon
2. Target parameters — Quantities that define the hypersurface, an analytic formulation that represents all possible translunar trajectories which satisfy the target objectives
3. Guidance parameters — The elements and orientation of the desired elliptical trajectory at TLI cut-off

The adequacy of a targeting procedure can be expressed in terms of a penalty function, that is, the penalty imposed on subsequent mission phases if the given targeting procedure is followed and the given target parameters and guidance parameters are used. For TLI, this penalty function could be the midcourse velocity change ΔV required to return the trajectory to the given target objectives for TLI.

Development of Targeting Techniques for the Apollo Missions

The development of the targeting procedures is discussed for four types of maneuvers performed during the Apollo lunar missions: (1) TLI, (2) TLMC maneuvers, (3) lunar-orbit insertion (LOI), and (4) return to earth (RTE), which includes not only aborts from the nominal mission but also the nominal transearth injection (TEI) maneuver and the transearth midcourse (TEMC) maneuvers. The development of targeting procedures for the TLMC, LOI, and RTE maneuvers was the responsibility of MSC personnel. Because of the nature of the Apollo Program, the TLI targeting procedure was developed jointly by MSC and NASA Marshall Space Flight Center (MSFC) personnel. Development of the target parameters and of the associated guidance equations was accomplished at MSFC. The real-time TLI targeting procedure and computer programs for the TLI targeting update were developed at MSC. Of these TLI developments, the real-time procedure and targeting-update computer programs are discussed in this report.

For each of the four maneuvers considered, a computer program was developed for use in real time to determine the guidance parameters for the maneuver. The historical development of these programs and of the related targeting procedures is discussed. To develop each program, it was necessary to consider the onboard-guidance capability, the mission objectives, the mission rules, the hardware constraints, the vehicle performance capabilities, the trajectory characteristics (principally geometry and energy), and the crew time line. Further development considerations were the current optimization and targeting techniques (state-of-the-art knowledge), the effect each maneuver would have on subsequent maneuvers, and the vulnerability of each maneuver to varying initial conditions. Because of the schedule imposed on the Apollo Program, the efforts in many of these areas were conducted simultaneously with the targeting development effort. Therefore, the progress of the targeting-technique development was directly related (1) to the developments occurring in other phases of the Apollo Program (e. g., the definition of guidance capability, hardware development, etc.), (2) to the development of analytical tools, and (3) to the definition of target objectives for each maneuver.

TRANSLUNAR-INJECTION TARGETING UPDATE

The TLI Targeting-Update Program was completed during the summer of 1968 and was first put into a real-time system for the Apollo 12 mission. The circumstances that would lead to the decision to update the targeting of the launch vehicle for the TLI maneuver had remained in doubt up to that time.

The development of the TLI Program was concurrent with the definition of guidance equations; the development of the mission profile; and the development of hardware, software, and the associated constraints. The TLI Program also depended heavily upon the Generalized Forward Iterator. Some inefficiencies occurred in TLI Program development; however, like many other developments in the Apollo Program, the TLI Program began with inexperienced personnel who faced a restrictive time line and the realization that this effort was also highly interdependent on other developments in the Apollo Program.

Pre-1967 Development

Interface between guidance equation development and targeting techniques. - From the outset of the Apollo Program until 1965, the launch vehicle guidance and targeting functions were not well defined. However, within a few years after the Apollo Program began, some detailed preliminary logic and equations for the command module computer (CMC) had been developed at the Massachusetts Institute of Technology (MIT).

Included among the programs available for the CMC was a set of guidance equations for TLI backup; that is, this guidance was to be used if the launch vehicle could not perform the steering for the TLI burn. Cross-product steering, which was the type of guidance for the CMC, used as target parameters a desired semimajor axis a and three position components of a target vector \hat{r}_T .

When the CMC equations became available, a significant effort was directed toward understanding, analyzing, and evaluating the equations and developing ground-based logic to support the targeting problem. Two approaches were taken to develop the targeting logic.

1. To iterate on the components of \vec{r}_T and on a to achieve an optimum TLI maneuver that would place the spacecraft on a free-return trajectory that had nominal perilune conditions

2. To iterate on ignition time and to vary the outbound trajectory in an attempt to fit the problem to the guidance capability

For the second approach, the TLI ΔV was not optimized within the cycle, but the overall effect of varying the set of the parameters on injection ΔV and midcourse ΔV was considered.

Studies were conducted for nominal and dispersed cases. A significant emphasis was placed on a study of dispersed cases. However, the dispersions used (for earth-orbit insertion) were for an open-loop Saturn IV (S-IV) guidance scheme and were greater than the dispersion characteristics of the closed-loop scheme used on the S-IVB for the lunar flights. The results of the study indicated a need for targeting update. However, in late 1965, dispersion characteristics became available for the S-IVB, and the study was performed again. This time, no targeting update was required in the 3σ case.

The main drawback of these initial studies was that only coplanar translunar injections were considered. Insufficient information was available about the targeting philosophy and the launch window design based on that philosophy.

Launch-vehicle guidance for TLI. - In the summer of 1965, an effort was begun to investigate the TLI guidance and targeting routines on board the launch vehicle. The following items were considered.

1. What would be involved in the TLI targeting update?
2. What were allowable 3σ deviations at earth-orbit insertion before the update had to be made?
3. Did any targeting-update capability exist in the launch vehicle; and, if so, what were the details?

In late 1965, an initial set of guidance equations was obtained. Within several months, the equations were programmed into a computer simulation at MSC. The targeting routine for TLI (the hypersurface eventually developed at MSFC) was preliminary and was not completely understood by MSC personnel for some time.

Initial development of the prototype real-time program. - Meanwhile, a prototype real-time program was under development by contractor personnel who used the analysis and development at MSC of the in-house version of the program. The TLI update problem in this prototype program was treated as an extension of the translunar midcourse problem. Like the midcourse, TLI was investigated by using a mission design program, a version of the Generalized Forward Iterator. The complicated logic for TLI update had two options: (1) to design TLI and accomplish a full-mission optimization or (2) to design TLI to inject the spacecraft into a free-return trajectory that had nominal perilune conditions. The logic had to have the capability to consider a change of the designated lunar landing site in real time. Finite burns and the required analytic calibration for the finite burns were taken into account. By June 1966, the prototype program was composed of the following necessary modules.

1. The integration package
2. The TLI guidance equations (launch vehicle and CMC)
3. The LOI guidance equations
4. The translunar midcourse correction package
5. The first-guess logic
6. The forward iterator
7. The TLI burn calibration
8. The Jacobi calibration
9. The LOI burn calibration
10. The CSM lunar-orbit plane change (LOPC) burn calibration
11. The TEI burn calibration

The first-guess package was designed only to place the trajectory of the spacecraft in moon reference and on the correct side of the moon. The launch vehicle TLI guidance equations, the Iterative Guidance Mode (IGM), had been programmed, as had all the burn calibrations. However, the TLI burn calibration had the capability for nominal cases only, and the other calibrations were to be nullified when CMC cleanup began and the spacecraft guidance laws changed for several of the burns.

Deletion of spacecraft TLI guidance. - Among the first changes that were made to the CMC guidance equations in early 1966 was the deletion of the TLI (\vec{r}_T, a) steering. A guidance interface still existed between the spacecraft and the launch vehicle; and, by mid-1966, an extensive effort was being conducted by MIT personnel to implement the TLI equations (IGM) into the CMC. Later in 1966, further deletions were made on CMC programs, and the IGM equations were among the first to be eliminated.

In summary, by the end of 1966 no change had occurred in the TLI update problem in the real-time prototype program. Full-mission optimization in earth parking orbit

was being considered. Because of a complicated setup of the iterator, many logical paths had to be followed in the program. The problem was sensitive and was heavily dependent on a combination of good first guesses and proper setup of the iterator. Much of the work that had been completed was inapplicable to the problem. Time had been spent at MIT and at MSC in the development and evaluation of the TLI (\vec{r}_T, a) back-up guidance equations, which were now useless. An extensive effort had been put forward at MIT to enter the IGM equations into the CMC, and digital and analog studies were well underway when the decision was made to take the IGM out of the CMC.

Program Development During Late 1966 and 1967

In 1966 and 1967, extensive development of the TLI targeting technique and of the real-time computer complex (RTCC) TLI Targeting-Update Program was accomplished. In late 1966, when no CMC program was available to back up TLI (although Lambert's targeting in conjunction with cross-product steering was being evaluated for this purpose at the time), interest increased concerning the capability to update the TLI targets on board the launch vehicle. The initial RTCC requirements for the TLI planning processor were to be due in the summer of 1967; therefore, it was necessary to determine as quickly as possible what capabilities it should contain. In April 1967, the onboard targeting-update capability was defined. Two targeting-update options were defined: (1) the hypersurface quantities were updated or (2) only the time of restart preparation and the desired cutoff ellipse were specified. During MSC-MSFC discussions, it was decided that the first capability was not required; therefore, the decision was made to support only the second targeting-update option.

Initial RTCC requirements. - In the summer of 1967, the first RTCC requirements were submitted to the contractor for the TLI planning processor. Although the significant decision not to allow a change of the desired lunar landing site in real time simplified the program, it remained somewhat cumbersome. A decision also was made to put into the RTCC TLI Targeting-Update Program the capability to update the CMC with Lambert's targeting for TLI.

In the initial definition of the TLI Targeting-Update Program, four options were available for real-time planning consideration.

1. Solve onboard targeting (hypersurface).
2. Optimize the full mission in earth parking orbit.
3. Maximize apogee altitude for the amount of S-IVB ΔV available.
4. Execute TLI to place the spacecraft in a trajectory with a specified apogee altitude.

Option 1 was indicative of what would happen if no targeting update was made. Option 2 optimized the service module fuel reserves for the full mission. Option 3 was made available for a nonnominal situation in which the S-IVB did not have adequate ΔV to perform the nominal TLI maneuver. Energy was maximized (in terms of apogee altitude), and a transfer maneuver possibly could be made from that trajectory to continue the nominal mission or to execute an alternate lunar mission. Option 4 was made

available for alternate earth-orbit missions and, more specifically, for the nominal earth-orbit lunar-landing-simulation mission (E-type mission) that was planned at the time.

All four options could be viewed for either a first or a second TLI opportunity. For each option, one pass to generate the spacecraft targeting for TLI (Lambert's targeting) was made through the IGM equations after the last iteration. As a measure of merit, a midcourse correction was computed for options 1 and 2; the midcourse was targeted back to the nominal nodal conditions.

The first set of requirements was rather complex. The IGM guidance equations had to be added to the program. The full-mission-optimization logic was complex and required extensive supportive analysis that would include analysis of the calibration-burn models. In 1967, the onboard (CMC) guidance equations were being changed constantly, because the requirements for onboard computer space were somewhat greater than the space available; thus, constant updates of the calibration-burn models were required. The one calibration on which work progressed steadily and by which good results were produced was that for an S-IVB-guided TLI. The major part of the work connected with this calibration was completed in 1967.

Significant simplification in RTCC requirements. - The results of a study conducted in late 1967 indicated that the full-mission optimization was not necessary to the TLI planning program. Thus, the second option in the program could be simplified. The study showed that nothing was saved by optimizing the mission while in earth orbit and that, by targeting to the nominal perilune altitude and latitude and for free return, no ΔV penalty would be incurred. A decision also was made to drop the TLI guidance program from the spacecraft computer.

These developments resulted in updated requirements for the TLI planning program. Full-mission optimization was eliminated as option 2; the new procedure was to target TLI to the nominal perilune conditions (altitude and latitude) and for free return. The procedure for generating spacecraft targets (Lambert's targeting) was removed from the program.

In summary, the program was at a relatively sound level in the winter of 1967 and 1968. Although the first set of programming requirements had been complicated, simplification of the second set had been made possible by the deletion of the full-mission optimization. The only inefficiency that occurred during this period resulted from changes to the guidance requirements in the CMC. However, at the end of this period, it appeared that onboard and ground-based logic was becoming stabilized.

Development of Current Programs

Further simplifications of the real-time program. - At the beginning of 1968, even though the TLI update problem was relatively simple, further simplifications were made. The IGM guidance equations were removed from the program. The retention of these equations was no longer necessary because of the deletion of the spacecraft TLI program.

Option 2 continued to be the cause of most of the problems. The first-guess package that had been developed for option 2 was empirical, and physical conditions could exist in which the first-guess logic would provide inaccurate first guesses to the iterator. The empirical simulation of the TLI burn used in all four options was hampered because the simulation was reliable only for the variations that had been considered for the parameters which it generated; the simulation did not have the capacity for very large plane changes that could occur in the iteration process. Eventually, workaround procedures were developed for all known possible sources of error; and, in the summer of 1968, a revised set of program requirements was sent to the contractor.

Impact of the Apollo 8 mission on program development. - In the fall of 1968, objectives for the Apollo 8 mission were announced. Because the RTCC would have to support a lunar mission within 4 months, it was decided to make the ground-based system in the RTCC as simple as possible and yet give it the capability to support the mission satisfactorily. Accordingly, the TLI Targeting-Update Program was not added to the system.

From the fall of 1968 to the summer of 1969, the TLI Targeting-Update Program was dormant. The program was not a part of the RTCC system for the Apollo 8, 10, and 11 missions. During the winter of 1968, a decision was made to eliminate the TLI Targeting-Update Program. In the summer of 1969, this decision was reversed, and the program was put into the RTCC system for the Apollo 12 mission.

Program on the RTCC system for the Apollo 12 mission. - In mid-1968, several discussions were held with MSFC personnel regarding the processor. Personnel at MSC assumed that MSFC personnel would verify the RTCC capability to update the launch vehicle and to perform a satisfactory TLI, based on the targeting update. Personnel at MSFC requested that definite guidelines be established in regard to when this update capability would be used. In subsequent discussions between personnel of the two centers, a conflict existed about establishing these guidelines; therefore, the verification effort was not begun.

Meanwhile, an outline of the requirements had been made at MSC for the targeting-update capability. It was generally agreed among MSC personnel that retargeting was required to ensure efficiency and to provide the broadest capability. The circumstances that were considered by MSC to warrant consideration of a real-time targeting update were as follows.

1. Provision for a real-time targeting-update capability in the presence of launch vehicle malfunctions or launch vehicle onboard-programing logic (or both) that would permit the achievement of earth orbit but with insufficient S-IVB propellant to achieve nominal TLI-guided cut-off

2. Provision for a real-time targeting capability in the presence of launch vehicle malfunctions or launch vehicle onboard programing logic (or both) that would permit the achievement of an earth parking orbit which would fall outside the capability of onboard targeting to compute a correct target or any target at all

3. Provision for a rapid (1 to 2 days) prelaunch targeting capability to accommodate last-minute changes to the mission or to the launch dates and to accommodate launch windows in excess of 3 or 4 days

4. Provision for the operational capability to accommodate real-time launch vehicle or spacecraft problems that could necessitate an injection attempt on the third opportunity

5. Provision for general operational real-time capability to accommodate any unknown or undefined launch vehicle or spacecraft problem that would prohibit the use of the normally computed launch vehicle TLI target

The program for the Apollo 12 mission had been changed little since the summer of 1968. A targeting-update command load was generated during the Systems Integration Test and sent to the S-IVB. A verification was received that the command load was accepted and ignition would be commanded at the correct time. The processor was available in real time for the Apollo 12 mission but was never called up, because the predicted TLI maneuver seemed reasonable.

Development of improved first-guess logic. - Because of the high perilune altitudes achieved when the hybrid trajectory profile was used, the first-guess package in option 2 yielded poor first guesses. Although this package was improved somewhat for the Apollo 12 mission, it was decided to develop a new analytic first-guess package that would be relatively insensitive to the trajectory profile flown. A task was assigned to one of the contractors for the development of the logic under the supervision of MSC personnel.

However, during the initial planning for the Apollo 13 mission, a decision was made to use a constant time of arrival at the moon as one of the target objectives for TLI. Because of this decision, the problems with the empirical first-guess package disappeared. If this philosophy is maintained for the remainder of the Apollo missions, it is probable that the empirical first-guess package will work well.

In summary, because of the time and effort that would be involved to incorporate the version from the contractor's task into the RTCC system, it was decided to keep the existing TLI program in the RTCC and not to implement the contractor-developed logic for either the Apollo 14 or the Apollo 15 mission. In fact, if the same philosophy of constant time of arrival is used for the remainder of the Apollo Spacecraft Program, the logic developed by the contractor personnel will probably not be needed for real-time use.

TRANSLUNAR MIDCOURSE

Early in the Apollo Program, lunar trajectories were found to be extremely sensitive to dispersions at TLI. Because of this sensitivity, it was improbable, from an operational viewpoint, that TLI would be performed accurately enough to achieve a trajectory which would permit a desirable LOI without the need to execute a TLMC maneuver. The RTCC midcourse-correction processor was designed to determine flight

maneuvers during translunar coast. Although an estimate of the probable deviations (3 σ) in TLI was derived from statistical studies, the RTCC processor had to be capable of computing an acceptable trajectory and target parameters from any possible cut-off vector at TLI.

Initial Requirements

It was realized that the TLMC maneuver would affect the time line of the entire mission and that complete optimization would entail recomputation of all major maneuvers to determine the best adaptive path (BAP) modes. If a severe dispersion at TLI precluded a nominal lunar mission, the midcourse processor would be used to compute alternate mission plans with better fuel reserves and to compute precision free-return trajectories (flyby modes) for use if a lunar-orbit mission was no longer desirable.

Optimization of the midcourse maneuver was of primary importance because spacecraft ΔV budgets were limited. Speed of computation was essential because the midcourse ΔV usually increased with delay time, and the evaluation of several alternative maneuvers might be required. Patched conic trajectories were used because many trajectories had to be computed to determine an optimum. It was assumed that, with a judiciously chosen value for the sphere of influence of the moon, the end conditions that were optimum for patched conic trajectories also would be optimum for a precision trajectory. The BAP modes of the midcourse processor were to be exercised only during the period from 5 to 20 hours after TLI.

Because of lead-time considerations, the existing mission-design program was used as the basic vehicle for the midcourse processor. However, the Generalized Forward Iterator was a general program with far more capability than the midcourse phase required. A much more specialized and regimented program was required for the real-time environment, and for this reason, a formal midcourse philosophy was needed.

The initial real-time midcourse program design philosophy was a product of the combined efforts of MSC and three contractors. The personnel in these groups specified and developed a set of mission profiles that the midcourse processor would compute. These profiles or options represented all the possible trajectory sequences that were considered feasible in mid-1967.

Computational Options of the Midcourse Processor

Detailed logic was developed for seven options in the midcourse processor during the initial program development.

Option 1 — nodal targeting. - For option 1 (X, Y, Z, and T targeting), a trajectory was computed from point A to point B. Option 1 was the simplest and, therefore, most foolproof of all options. At the midcourse position, the maneuver was computed to determine a trajectory that would bring the spacecraft to the nominal location (X, Y, Z) of LOI at the nominal time T. This option was applicable to any mission scheme, if a set of target parameters was known. The resulting trajectory was a free return only if the nodal targets were associated with a free-return trajectory, based upon the position of the spacecraft at maneuver time.

Option 2 — fixed lunar parking orbit, free return. - Because Apollo Program planning was committed to free-return trajectories, the midcourse processor had to have the capability for computing a free-return trajectory that would intercept the nominal lunar parking orbit. This specification resulted in option 2, the fixed-orbit, free-return BAP. While retaining the nominal lunar-orbit orientation, this option reoptimized a complete free-return mission profile that included LOI, an LOPC to enable the command and service module (CSM) to pass over the lunar landing site a second time, and TEI maneuvers.

Option 3 — free lunar parking orbit, free return. - Concern over dwindling fuel reserves because of continued weight growth of the spacecraft resulted in the inclusion of option 3, the free-orbit, free-return BAP. Because of the launch-window-design philosophy, a single lunar-landing-site approach azimuth, usable over a 3-month period, was to be selected on the basis of positive fuel reserves. Therefore, on a specific launch day and with a TLI dispersion, the nominal azimuth was not necessarily optimum. To provide for this contingency, option 3 was devised by retaining the free-return aspect of option 2 but freeing the azimuth at the lunar landing site. Full optimization determined a new approach azimuth that resulted in slightly less propellant being used than in the solution computed by option 2.

Options 4 and 5 — nonfree return. - Options 4 and 5 represented a logical extension of the ideas that produced option 3. If TLI dispersions were great enough to make a free-return mission too costly in terms of ΔV , a nonfree return (NFR) possibly would permit continuation of the mission. Relaxation of the free-return trajectories was not considered for the first lunar flights. However, the lunar module descent engine could be used, within limits, as a backup (descent propulsion system abort) to return the spacecraft safely to earth. Because of this safety margin, limited NFR trajectories became an actual alternative in real time.

Option 4 (fixed-orbit NFR BAP) was designed to provide the midcourse processor with any capability that might be required for the use of NFR trajectories while retaining the nominal approach azimuth to the lunar landing site. Because translunar flight time was not uniquely determined on an NFR trajectory, a polynomial $\delta(\Delta T)$ was developed to predict the optimum translunar trip time during the first 20 hours of translunar coast (based upon the position and energy of the spacecraft and the current earth-moon geometry).

Option 5, the free-azimuth version of option 4, had so much latitude that it would not reliably optimize to completion. Moreover, if a zero ΔV LOPC maneuver was encountered, a nonoptimum solution would often be returned. To overcome this difficulty, the user of option 5 was allowed to specify by input (manual-entry device) whether to allow a range of approach azimuths or to compute a discrete, nonnominal azimuth. By using a series of discrete azimuth solutions in conjunction with a midcourse Trade-Off (TO) Display, the dependence of end-of-mission fuel reserves upon the approach azimuth could be determined. Option 5 was included in the program specifications but was recommended for use in real-time analysis rather than for mission support.

Options 6 and 7 — lunar flybys. - Options 6 and 7 (lunar flybys) completed the original set of mission profiles. The intent was to provide computational support for alternate missions in the event of a badly dispersed TLI. It was in this aspect that questions arose concerning the prevailing assumptions and specifications for optimization. Option 6 was supposed to be an optimized lunar flyby passing through the nominal perilune latitude and altitude, whereas option 7 was to retain the constraint on perilune latitude, and the user was to specify perilune altitude by manual input. The prevailing assumptions and constraints were not unreasonable for large midcourse maneuvers (a ΔV of approximately 1500 ft/sec) executed early in translunar coast. Targets that were determined by conic optimization yielded midcourse maneuvers on integrated trajectories that were nonoptimum by approximately 35 ft/sec; for extensive midcourse maneuvers, this error could be considered acceptable. However, for small midcourse maneuvers such as those that occurred during the Apollo 8 mission, the differences between conic and integrated trajectories caused the small dispersion to be overwhelmed and thus invalidated this entire approach. In mid-1967, it was known that, if dispersions occurred, a perilune altitude other than nominal could result in a less expensive lunar flyby. Therefore, option 7 was created in the hope that repeated use of the option might permit the user to determine empirically an optimum lunar flyby during real time.

Adapting to the Real-Time Environment

The complexity of the various mission profiles and the real-time requirement of speed and reliability were considered indications of the need for program automation. The idea of "wiring the man out" to reduce human error was a significant factor in the construction of the program supervisor logic. Each option was to be executed as automatically as possible from stored data. However, the forward iterator needed initial values, or first guesses, to begin solving problems. These values were critical and varied significantly with different launch days and even with different launch azimuths and TLI opportunities. The conflict of varying initial values and stored data was solved primarily by contractor personnel with the concept of the Data Table.

The Data Table was designed to store the independent variables in such a way that the following conditions existed.

1. The nominal values could be loaded preflight.
2. The nominal values could be updated by midcourse computations.
3. All values could be manually overridden.

The Preflight Data Table was used in program testing. Because a table was required for each possible time of launch, 10 distinct Data Tables existed for each launch day. Preflight Data Tables were generated for launch azimuths of 72° , 81° , 90° , 99° , and 108° for both first and second TLI opportunities. When a specific launch azimuth was selected, the appropriate Data Table was constructed by interpolation. This Data Table was called Data Table 1.

The midcourse processor, therefore, started each mission with an interpolated table. When a BAP was computed, the processor would create an updated table from the associated values of the independent variables. The midcourse processor could store (in the midcourse TO Display) as many as six computed mission profiles. Each BAP that was displayed had the implicit capability to produce an updated table. Such a table could be created and displayed by requesting Data Table 2. This second table was experimental in that a certain amount of mission planning was allowed to continue without affecting the rest of the midcourse system. Any succeeding midcourse computation could use either the first or the second table for initial values. The second table, based upon precision propagation, was usually preferable, because the first table held only conic data.

When a computed midcourse BAP maneuver was actually executed (or transferred to the Mission-Plan Table), the independent variables associated with the maneuver were loaded into Data Table 3. All succeeding midcourse computations used Data Table 3, unless otherwise specified. Independent variables associated with the next executed BAP maneuvers were loaded into the fourth and fifth tables. If still another maneuver was executed, the table from that BAP went into the fifth table. The old data in the fifth table went into the fourth table, and the fourth into the third; and the old data in the third table were deleted. By this procedure, the execution of any number of BAP maneuvers was allowed, and an adequate maneuver history was retained.

Limitations of Conic Optimization

Another difficult problem in the midcourse processor stemmed from the relationship between conic- and integrated-trajectory propagation. For computational speed, a conic-trajectory computer was used for all of the options during optimization. When optimization was complete, an integrated trajectory was computed that satisfied the same end conditions at the moon. The midcourse maneuvers obtained with the conic and precision trajectories were considerably different. To decrease the number of iterations required to converge precision trajectories, a polynomial $\delta(\Delta V)$ was developed for application to the conic value for the scalar velocity at midcourse. Contractor personnel later suggested that computations begin with the desired end conditions at the moon and converge to the midcourse position with precision propagation to overcome this same difficulty. When this scheme was incorporated into the program, the results were even better; thus, the $\delta(\Delta V)$ polynomial was discarded.

Because the midcourse maneuver that was predicted during optimization was considerably different from the maneuver derived by using precision trajectories, the solutions were not fully optimized. Near the end of 1967, personnel of one of the contractors realized the inadequacy of the specifications for optimizing lunar flybys, but the problem could not be solved at that time because of insufficient understanding of flybys.

Apollo 8 Mission — Lunar Flybys and the Vector Offset Techniques

Considerable effort and time were expended to improve the conic program by reducing the program error. Work was completed on lunar-sphere-size studies, Jacobi calibrations, and hybrid patched conic trajectories. When the Apollo 8 mission was scheduled as a CSM solo lunar mission, optimum lunar flybys were required to be computed from near-nominal trajectories at any time during translunar coast. This new requirement was met by the vector offset method that was developed at MSC during the activity which preceded the Apollo 8 mission.

By use of the standard conic- and integrated-propagation routines, the supervisor logic was structured to measure the difference in midcourse maneuvers between comparable conic and precision trajectories. This difference was applied as a velocity bias, or offset, to the premidcourse state during conic optimization. Thus, the integrated midcourse maneuver could be optimized while a conic trajectory computer was being used.

Application of the vector offset to the flyby optimization opened up a new field of lunar trajectories as possible solutions to the problem. These trajectories led to a better understanding of the problem and resulted in a complete revision of the original flyby specifications. New flyby options, 8, 9a, and 9b, provided a variety of flyby capabilities. By use of option 9a, a flyby was possible that used the true minimum ΔV to return to earth within a range of inclinations of return and perilune altitudes. When option 9b was employed, the optimum lunar flyby for a specified inclination and range of perilune altitudes was achieved. By use of the computational support provided by option 9a, the free-return landing point could be moved from any place on earth into deep water, for an additional ΔV of less than 50 ft/sec. The use of option 8 resulted in a lunar flyby to a specified inclination of free-return and perilune altitude. Although option 8 did not involve optimization, the vector offset was used to compute accurate first guesses for the integrated maneuver. Because of a lack of lead time, the new flyby options were not incorporated into the Apollo 8 RTCC midcourse processor, but these options were made available in the Real-Time Auxiliary Computing Facility and were exercised throughout translunar coast during the Apollo 8 mission. All modes of the midcourse processor were required to function throughout translunar coast.

Vector Offset Applied to NFR Options

In early 1968, the NFR options, as specified, were found to be nonoptimum for small midcourse maneuvers because of the ubiquitous difference between patched conic and precision propagation. Application of a velocity bias at the midcourse point could not completely eliminate the problem, because, unlike the case of free-return trajectories, translunar flight time on an NFR trajectory was not uniquely determined by the nodal position. The application of a velocity offset at both ends of the translunar trajectory during conic optimization and the use of flight time to terminate each trajectory comprised the solution. A consequence of the solution was coupled optimization of both the midcourse and the LOI maneuvers into a close approximation of the actual maneuvers. The principal values of NFR trajectories were recognized to be the fuel savings and translunar-trip-time flexibility when these trajectories were used in conjunction with the hybrid mission profile.

Apollo 10 and 11 Missions — Vector Offset Incorporated into Free-Return BAP Options

The decision to use the same lunar orbit orientation on the Apollo 10 flight as on the Apollo 11 flight was made too late to retarget TLI. The RTCC midcourse processor would be used to compute midcourse maneuvers during translunar coast in order to permit an LOI into the desired lunar orbit. The mismatch between the nominal translunar trajectory and the lunar-orbit orientation showed that, as originally specified, the free-return BAP options would yield nonoptimum answers in the region approximately 20 earth radii from the moon. Two alternatives were available: (1) change of the weighting scheme to place more importance on the plane change at LOI (a form of symptomatic treatment) or (2) incorporation of the vector offset technique. The latter was selected and resulted in the program that was also used for the first manned lunar landing.

Ensuring RTCC Reliability

The most difficult and prolonged task of all was the effort to make the program completely reliable. For the real-time situation, a program was required that would not fail. The RTCC midcourse was required to optimize the smallest of maneuvers and still retain the capability to correct dispersions for which corrective maneuvers of several thousands of feet per second were needed.

Because of the nature of iterative techniques that involved several independent variables, the Generalized Forward Iterator required an exhaustive study to determine the most reliable independent-variable step sizes and weighting schemes. Fine tuning of these parameters for a specific launch date or maneuver time could cause disastrous results in cases for which a different geometry was encountered. In addition to studies that were conducted to determine the proper step sizes and weighting schemes, critical trajectory routines such as Patch and LOPC that usually worked well were reconstructed to eliminate all known problems. At times, this reengineering entailed yet another weight and step-size study.

In summary, the RTCC midcourse processor gradually evolved over a period of years as a result of constantly changing requirements. Key ideas and programing support were contributed by personnel from various organizations. Many major developments were actively pursued, only to be abandoned or to remain unused after being fully developed. Appreciation of the problem and of the behavior of the iterator resulted in specifications involving a sequence of steps that led to a final optimization computation instead of a single all-out optimization. The resultant program was more complex but afforded greater reliability. Perhaps the greatest progress was in human understanding of the problem and of the iterator used to solve it.

LUNAR-ORBIT INSERTION

Work on the real-time LOI Targeting Program proceeded in four phases. The first phase (pre-1967) involved learning about the available guidance equations and the inherent capabilities of the equations. The second phase (spring 1967) was the first

attempt at a targeting program. The problem of developing a real-time targeting procedure was assumed to have been solved by an iterative technique in which the burn was simulated by integration through the guidance equations; therefore, it was assumed that the program had only to be built. This attempt was unsuccessful because the LOI problem was more complex than was originally conceived. In the third phase (1967 to 1968), the problem, after closer examination, was better understood, and a design philosophy and program design were developed that should have been workable. An iterative technique and integration through the guidance equations were used. However, before this program was built, the fourth phase was begun, in which operational simplifications were made to the LOI burn. The most important simplification permitted the LOI burn to be simulated by an impulse; therefore, the program actually used for the Apollo lunar missions was drastically changed. The philosophy that was developed in the third phase of the program remained the same, but the method of implementing the program was changed.

Pre-1967 Development

Interface with guidance-equation development. - In 1964, MIT personnel developed a steering law (cross-product steering) for the spacecraft, which required values for a velocity-to-be-gained vector. For the various CSM maneuvers, several computer programs were used to compute the values of this vector. A circularization guidance that was to be used for LOI was designed to provide a circular orbit at burnout, which occurred whenever the spacecraft position and velocity in the burn had an osculating eccentricity of zero. Also, with this guidance law, the spacecraft would pass through a given inertial position. During the early work on the LOI targeting problem, the intention was to gain knowledge of the capabilities of circularization guidance. This work was also necessary for the mission-planning trajectory scans; no effort was directed toward the specific study of the LOI targeting procedures. Because circularization guidance could result only in a circular orbit at burnout, no other profiles were studied.

Objectives of initial studies. - The initial studies were completed through in-house efforts and tasks performed by prime contractors. The objectives of these studies were twofold. One objective was to determine which guidance variables controlled each trajectory parameter. Because no closed-form analytical simulation of the burn was found, an iterative approach was used to determine guidance variable values (e. g., ignition time) to obtain specified orbits after the burn. The necessity of iterating was eliminated only when the LOI burn became so restricted by operational constraints that the burn could be simulated by an impulse. The second objective was to develop empirical equations to compute various guidance variables and to develop polynomial simulations of the LOI burn. These developments were primarily for use in the mission-planning scan work to avoid the lengthy computer-run times associated with iteration through the guidance equations.

Results and observations. - Several usable ideas were obtained from evaluation of the observations. A good understanding of circularization guidance and some understanding of cross-product steering were acquired. Empirical equations for guidance variables and polynomial burn simulations were developed, and the major LOI trajectory problems that would have to be processed in the real-time targeting-update program were observed. However, no detailed knowledge existed concerning the LOI trajectory problems (e. g., the effect of trajectory dispersions on the node of two orbits that are

nearly coplanar); only guidance capabilities were known. The iterative technique was solidified as the solution to the LOI targeting problem, just as it had been for the midcourse-correction problem.

Several observations can be made about this early effort. Possible real-time targeting problems had appeared but were ignored. The LOI was not regarded as a serious problem because the RTE and midcourse problems were so complex compared with LOI that LOI was deemed relatively unimportant. Nevertheless, it was a mistake not to have had at least one person working full time on the LOI problem. A tendency developed to take the approach of "do it like the midcourse" or "the iteration can solve the problem" and to ignore the problems that had occurred. If this mistaken approach had continued, a workable LOI targeting program probably could not have been developed in the available time without a large effort being expended; however, the delay caused by the spacecraft fire in January 1967 provided additional time for work on the problem. Consequently, real-time capabilities and mission-planning freedom were not hampered.

The probable reason for these mistakes was that the scope of the early work was too limited. The analysis of the guidance equations was necessary, but the development of the LOI targeting procedures should have been studied independently of the specific guidance problems. Profile changes, guidance changes, and trajectory problems should have been considered to gain an understanding of the real-time targeting problem so that the necessary engineering trade-offs could be made and a real-time targeting program developed.

First LOI Targeting Program

Considerations. - The development of the real-time LOI Targeting Program was begun in January 1967. A prototype set of RTCC computer programs for Apollo mission planning had been under development during 1964 and 1965. The prototype RTCC logic contained an LOI Targeting Program that was an iteration on ignition time and could not have handled dispersions. Little difficulty was expected in developing the program, because it was assumed that only proper initialization of the iterator was needed; that is, the problem was assumed to be solved already. However, several events that influenced the program were occurring at this time. Some operational considerations were being advanced, mainly the problem of onboard burn monitoring, the solution of which later led to the elliptical lunar orbit after LOI and to the fixed-attitude LOI burn.

Impact of initial operational considerations. - For the E-type mission, a simulated LOI burn into an elliptical orbit was performed. Because circularization guidance could not be used for this maneuver and because the guidances that could be used had been eliminated from the onboard computer to gain computer storage, a rendezvous guidance (Lambert's guidance) was evaluated for this maneuver. In this guidance, the velocity-to-be-gained vector was determined from Lambert's problem (given a time of flight and a target vector). It was decided to include additional guidance in the LOI Targeting Program to handle elliptical orbits after LOI. The development of the program logic would not be delayed for this guidance; however, the guidance logic would be added later if it could not be included in the initial program specifications. External ΔV guidance (eventually used for all CSM burns) was available but was not considered because of the concern that this guidance might be too sensitive to dispersions.

As the LOI Targeting Program developed, problems began to arise more frequently. The assumption that the problem was already solved inhibited complete examination of the problem, determination of the real nature of the problem (in more specific terms than the generality, "target LOI"), and determination of the necessary engineering trade-offs. As a result, this first targeting program was inferior. Many logic paths existed, each with an individual iterator setup. The run time was estimated to be long; and, because the simulations were available, an attempt was made to use the polynomial burn simulations developed in earlier work. This approach made the program highly inflexible, because the polynomials were tied to a specific profile and vehicle performance. Because of the problem with the program run time, consideration was given to expanding the polynomials so that they would have the capability to handle elliptical lunar orbits.

Another aspect of the problem was that personnel working on the program were inexperienced in this type of work. A plan of attack and the directions the program development should take had not been established. However, this first attempt at a program resulted in valuable experience in real-time-program design.

In the first attempt at the development of an LOI program, approaches to the problem were broadened. A better understanding of the iterator and of cross-product steering (independent of the velocity-to-be-gained vector computation) was deemed necessary. A critical realization was that situations occur when all the desired end conditions cannot be attained (i. e., a physically insoluble problem) because the approach hyperbola is fixed. Therefore, one or more end conditions must be relaxed. However, the best maneuver that could be targeted would be as close as possible to the relaxed end conditions. It was also realized that, with the nature of cross-product steering and vehicle-performance capabilities, a maximum allowable ΔV for LOI would have to be specified as an external input to the program. These observations were the result of a closer view of the problem than had been taken in the earlier work. However, by early 1967, a workable program was still a long way from completion.

Second LOI Targeting Program

Need for increased flexibility. - As the time for manned lunar flights approached, operational considerations were becoming more and more important. For this reason and because of the dissatisfaction with the first efforts at developing a program, a decision was made to take a fresh look at the LOI problem.

Many perturbations occurred in the development of the LOI Targeting Program, which was originally designed to support a nominal 80-nautical-mile circular lunar parking orbit. The circularization logic was deleted from the onboard software. In addition, concern over degraded lunar module performance because of increased lunar module weight and the LOI monitoring problem resulted in reconsideration of elliptical lunar parking orbits rather than circular orbits, of lower altitude orbits, and of fixed-attitude steering rather than Lambert's steering. Because of these uncertainties, a decision was made to redevelop this program to provide more flexibility.

Design philosophy. - Reconsideration of the development of an LOI Targeting Program resulted in the development of a good design philosophy and the necessary technical understanding to implement it. The design philosophy that was developed by late

1967 was carried over to the final program and primarily was responsible for the quick program development (approximately 1 month). This philosophy consisted of three parts.

The first part of the design philosophy was carried over from the early program development efforts and included the idea of relaxing end conditions. Initially, trade-offs had been made between obtaining the desired landing site and obtaining the desired orbit shape. In this second effort, the following priority was set: the desired orbit shape was to be obtained within the maximum allowable LOI ΔV constraints, and then the desired landing site was to be obtained if the orbit shape had been obtained. The reason was that the orbital geometry sometimes made passage over the landing site either impossible or obviously unreasonable, even with reasonably small dispersions in the lunar approach hyperbola, whereas only unreasonably large dispersions of the approach hyperbola would prevent achievement of the orbit shape. During the first efforts, either the landing-site latitude or the miss distance at the landing site had been used as the iterator's dependent variable to obtain a trajectory that would have been as close to the landing site as possible, within landing-site azimuth constraints. In the second effort, because it was realized that what was desired was an orbit plane (or planes, if a range of azimuths at the landing site was acceptable) that passed over the landing site, the landing-site variables such as azimuth (or a range of azimuths), latitude, and longitude could therefore be replaced by a single variable, a wedge angle. Thus, it was necessary only that this wedge angle at the landing site be driven as near as possible to zero during the iterations, because the iterator could optimize in only one variable (i. e., drive the variable as near as possible to a desired result). Therefore, the concept of the wedge angle was one of the most critical realizations, for the landing site could now be attained by optimizing on only one variable (wedge angle). Optimization on the orbit shape was accomplished by fixing apolune altitude and optimizing perilune altitude (for elliptical orbits) or by optimizing circular altitude (for circular orbits).

The second part of the design philosophy involved the elimination of errors that resulted from making the lunar-orbit shape more important than the landing site. The solution was to provide the flight controller with more information than that required to evaluate only the planned maneuver. Therefore, the flight controller would have the capability to obtain a more acceptable maneuver or to make trade-offs. For example, the flight controller was provided with information about the intersection between the approach hyperbola and the two planes that passed over the landing site with the specified two extreme azimuths.

The third part of the design philosophy was to make the program general by specifying as many degrees of freedom as possible to be left as inputs (e. g., the apolune and perilune altitudes after LOI). This philosophy allowed the flight controller to use available information to compute other maneuvers if desired; this approach was also required because of uncertainties in the lunar-orbit profile and in the guidance.

The second and third parts of the philosophy can be summarized by the statement that the program was designed to support automatically the known LOI options, but enough additional information was computed and displayed, and enough inputs were provided, to handle the unexpected. It was hoped that the program would then be stable and relatively immune to perturbations or that at least the program would handle changes in the lunar-orbit profile procedurally until the program could be modified to handle these new profiles automatically.

Engineering trade-offs. - To develop the design philosophy, a complete understanding of the trajectory and operational aspects of the problem was needed. With this understanding, the necessary engineering trade-offs could be made in order to implement the philosophy. The following characteristics of the program were considered.

1. Iteration through the guidance equations could be used.
2. The lunar-orbit plane and shape could be redefined by input.
3. Lambert's targeting and external ΔV guidance could be used.
4. Orbit shape could have priority over the landing site.
5. No trade-off could be made between LOI ΔV and any in-orbit ΔV .
6. Often, the ΔV remaining could not be optimized.
7. No in-orbit maneuvers could be used to attain targeting objectives.

In addition, polynomial burn simulations were to be used; however, because of the inflexibility of the polynomials, it was decided later not to use them.

Work was being completed to provide the necessary background for building the program. Lambert's guidance was thoroughly evaluated to determine which variables controlled each trajectory parameter and which variables should be used in the iteration. Because the end conditions were sometimes unattainable, an investigation of the optimization mode of the iterator was attempted. The important problem of lengthy computer run time was investigated by trying methods of further integration through the guidance (e. g., larger step sizes). A computer program was to be built to allow evaluation of some of the ideas. However, development of the program was slowed by the contractor personnel's lack of experience with the iterator and lack of understanding of the engineering, and by the lack of detailed direction and programming specifications by MSC. By the time this computer program was available, the final real-time program was already in use (with an impulsive burn simulation and no iterations); consequently, the contractor's work was never used.

In summary, a workable computer program evolved as a result of the following four developments.

1. The operational inputs were obtained.
2. A good understanding of the problem was achieved.
3. A new approach in thinking was taken.
4. A design philosophy consistent with the first two developments was adopted.

Most of the early work was not useful. Empirical equations, burn polynomials, and circularization-guidance knowledge eventually were discarded. However, the knowledge of cross-product steering and of LOI trajectory problems was useful. The second LOI Targeting Program was never advanced to the engineering stage; that is,

the computer program was never built so that the theory could be tested. Much time would have been required to develop a real-time computer program based on the theory. However, it was thought that a good basis had been established and that the second LOI Targeting Program had the capability to handle all LOI targeting changes that were likely to arise. This approach would not have been desirable if the problem were more constrained, but, because of the nature of the guidance, the approach was required.

Final Program

Operational inputs and simplifications. - At this point, a major operational input was obtained. To make the onboard attitude monitoring easier, a fixed-attitude maneuver was selected for LOI. The performance penalties of a fixed-attitude burn over a variable-attitude burn were negligible. Also, the decision was made to use an elliptical orbit after LOI to ease the overburn-monitoring problem. Because of these constraints, some major simplifications could be made.

In one of the studies conducted during the second effort at developing a program, a very important discovery was that a fixed-attitude burn resulted in a postmaneuver orbit of which the altitude at the node between the premaneuver and postmaneuver planes was equal to the premaneuver orbit altitude at that point. Alteration of the burn ΔV or attitudes (or both) would not achieve a different result. This situation was unforeseen, and considerable time was spent in an effort to understand the problem. The discovery was important because, when a fixed-attitude burn was selected for monitoring reasons, it was no longer necessary to iterate through the guidance equations. Because the LOI burn could be simulated by an impulse, the lengthy computer run time and the possible unreliability associated with an iterative solution to the LOI targeting problem were eliminated. The only iteration necessary would be to determine the guidance parameters needed to burn into the impulsively defined orbit, and this procedure would be no problem because of the work that had been done previously for many different hyperbolas and lunar-orbit geometries.

It might have been possible that approaching the problem initially from a general viewpoint, that is, concentrating on guidance and trajectories in general rather than on a particular guidance (circularization), would have shown that a fixed-attitude burn resulted in no nodal altitude difference. If the desirability of the fixed-attitude burn had been revealed, it is probable that, early in the program, the fixed-attitude LOI burn could have been adopted and considerable work avoided. To justify the fixed-attitude burn, the operational simplifications would have had to have been reached at an earlier date.

Use of the fixed-attitude burn eliminated the capability of targeting out planned nodal-altitude differences. However, because of the elliptical lunar orbit, this capability could still be achieved when the nodal altitude increased from the planned value as a result of dispersions, because the ellipse line of apsides could be rotated until the nodal altitudes matched.

The second LOI program (which was based on use of the iterator) was not the best for particular constraints, but it would be acceptable for a wide range of constraints. Because of these new operational constraints, drastic changes were made in the LOI Targeting Program. The philosophy remained the same; the changes made were in the method of implementation.

Construction of the targeting program. - Relaxation of the end conditions was maintained because physically insoluble problems could occur. However, instead of computing only one solution (maneuver) that could attain the orbit shape within a given ΔV and then could obtain the landing site, 10 solutions were computed with various end conditions relaxed (perilune altitude, desired azimuth, wedge angle, etc.). These 10 solutions improved the program because several maneuvers were now available, and the flight controller could select the best maneuver, to the extent that the best could be determined by the mission conditions at that time. It was possible to compute the 10 solutions because of the computational speed that resulted from the impulsive burn simulation.

Relaxation of the requirement for passing directly over the landing site was still accomplished by the wedge-angle concept. Minimizing this wedge angle within a ΔV constraint was the means of coming as close as possible to the landing site. It was possible to compute the wedge angle at the landing site from the LOI impulsive point because solutions with a range of azimuths over the landing site have angular momentum vectors that very nearly lie in a single plane. After LOI, the angular momentum vectors of the planes that would pass over the landing site could be assumed to lie in the plane formed by the angular momentum vectors of the orbits (propagated back to LOI) associated with the two extreme azimuths over the landing site. With the use of this concept, the computation of the various LOI solutions was straightforward and simple.

The 10 solutions fulfilled the requirements of the second part of the philosophy, providing the flight controller with information to make trade-offs and to use the inputs intelligently. Most of the degrees of freedom to the problem were specified as inputs.

Targeting program use. - The desire for a circular orbit at the time of the constant delta height (CDH) maneuver during rendezvous on the Apollo 11 and 12 missions illustrated how well this philosophy had worked. No automatically computed solution existed to establish a second maneuver to obtain a circular orbit at some future point after LOI, but a lunar-orbit perilune altitude could be determined that would allow LOI-2 to result in an orbit which would be perturbed so that it was circular at CDH. Determination of the lunar-orbit perilune altitude required knowledge of the nodal altitude and of the true anomaly on the hyperbola (which were computed by the program and displayed) and availability of the target perilune altitude as a program input. Therefore, in real time, the correct target perilune altitude would be determined from preflight-computed graphs after one run with the nominal perilune altitude. Thus, a new program requirement could be fulfilled by preflight work and real-time evaluation, without a change in the real-time program.

Summary and observations. - Program deficiencies did develop, however. Not all degrees of freedom were specified as inputs, and one particularly serious omission was the control of one of the two possible ways the line of apsides of the post-LOI ellipse could rotate. Some awkward procedures were used in the attempts to overcome this program deficiency. The omissions resulted in program changes. Some of the "do it like the midcourse" philosophy still existed, especially regarding the computation of the ΔV remaining after TEI. Some difficulty resulted from the failure to view this portion of the program as more than a computation that had to be performed and then added to the display data after computing the impulsive solutions. Also, some of the procedures that were performed "like the midcourse" could have been accomplished

by a better method. For example, the lunar orbit was propagated by using conic trajectories with nodal regression, primarily because the same method was used for the midcourse program.

After several missions, it was found that the problem had been oversupplied with solutions for a range of approach azimuths to the landing site, because it became apparent that ground personnel were very reluctant to use an approach path other than the one for which the flight crew had trained and for which the lunar-orbit propagation had been verified. Although it was desirable, the computation of ΔV remaining after TEI was determined to be unnecessary. However, because of the situation at the time the program was designed (i. e., that landing-site azimuth could easily change in real time), justification for excluding the range-of-azimuth capability would have been difficult. When the landing-site azimuth changed, the ΔV remaining after TEI could also change significantly; thus, justification for excluding the ΔV remaining after TEI would have been difficult also.

The maneuver targets were computed by the Maneuver Planning Table (MPT), which iterated on guidance parameters to give a burn that attained the impulsively defined lunar orbit. Originally, the guidance parameters were to be computed in the LOI Targeting Program to eliminate the MPT interface. This type of computation was not used because the program could be put into the RTCC more quickly with the MPT interface. No problems arose as a result of this interface; therefore, the correct decision was not to compute the guidance parameters in the LOI Targeting Program. However, some inputs to the MPT were based on knowledge of the LOI geometry. To ensure that there would be no trouble in the computation of guidance parameters in the MPT, a set of controllable variables was defined in the iteration for guidance parameters.

No inputs were available from the Lunar Orbiter Program. An attempt was made to determine the Lunar Orbiter procedures for LOI in the fall of 1967, but this study was unsuccessful.

Although it was a mistake to delay the start of the program development for so long (until early 1967), the delay probably resulted in the development of a much better program. Had a program been working before the decision was made to use the fixed-attitude burn, it would have been difficult (although not impossible) to change the program. Consequently, the program-development delay probably resulted in a better real-time tool.

Three major observations can be based on this experience.

1. Operational inputs are important because operational simplifications may result in drastic changes to the targeting program.
2. Program generality and flexibility are necessary because program requirements and operational constraints can change.
3. A thorough understanding of the problem is required for the development of a design philosophy consistent with the trajectory problems and the desired functions of the program and for an approach to the problem with an outlook that is not dependent upon previous methods for the solution of other related difficulties.

RETURN TO EARTH

The RTE Abort Program was a real-time support processor through which maneuvers could be computed to place a spacecraft on a trajectory that would return the spacecraft to the earth. These maneuvers might have been determined for any state vector contained in cislunar space. Although the program name implied that abort or nonnominal maneuvers were generated, the program also could compute the nominally planned TEI and TEMC maneuvers.

Current Program Configuration

The RTE Abort Program presented solutions on one of three real-time output displays: an analog display (the TO Display) and two tabular displays called the Abort-Scan Table (AST) and the Return-to-Earth Digital (RTED) Display. The displays differed in (1) the amount of information given for each solution, (2) the number of solutions presented, and (3) the precision of the solutions presented.

Trade-Off Display and solutions. - The TO Display provided solutions that were characterized by impulsive maneuvers, by conic (analytic) or patched conic coasts from maneuver to entry, and by a curve-fit entry simulation which would produce the landing point. This display presented a continuum of solutions for a range of maneuver and landing times on the premaneuver trajectory. Only the solutions that would return the spacecraft to a specified landing site were produced by the TO Display. Two types of landing sites were available. The primary target point (PTP) site was a specific latitude-longitude point on the surface of the earth. The alternate target point (ATP) site was a predominantly north-south line on the surface of the earth defined by as many as five latitude-longitude points. For both solution types, the TO Display presented the maneuver ΔV (for different landing times) plotted as a function of maneuver time. In addition, miss distance to the PTP was given for the PTP mode, whereas latitude of landing was given for the ATP mode. Based upon the availability of solutions within the constraints and maneuver times, the TO Display might contain a single solution or as many as 240 basic solutions.

Abort-Scan Table Display and solutions. - The AST displayed up to seven discrete solutions that were characterized by impulsive maneuvers, by precision-integrated coast to entry, and by curve-fit entry simulation. For each solution, 17 parameters, including those on the TO Display, were tabulated. Eight solution modes were available in the AST. At a specified maneuver time, the AST could produce ATP and PTP solutions that corresponded to the solutions of the TO Display. A fuel-critical, unspecified-area (FCUA) mode at a discrete maneuver time was available. A time-critical, unspecified-area (TCUA) mode and a specialized FCUA mode called the extreme fuel-critical, unspecified-area (EFCUA) mode were available for earth reference. For moon reference, the three discrete modes (ATP, PTP, and FCUA) also had a search-mode counterpart. For the search modes, the solution was the maneuver within the constraints that required the lowest maneuver ΔV over a range of maneuver times.

Return-to-Earth Digital Display and solutions. - The last display, the RTED, contained only two solutions. These solutions were the most precise in that they were totally integrated from the finite thrust maneuver to the guided atmospheric entry. More

information describing each solution was also displayed; 55 parameters were listed. The RTED Display generated a solution either from one of the solutions currently stored in the AST Display or from a solution defined manually by the flight controller. The RTED Display solutions were available for transfer to other processors, and a spacecraft target load could be generated for either solution.

Characteristics of display usage. - Two general characteristics of the RTE Displays, in the progression from the TO Display to the AST Display and then to the RTED Display, were that (1) fewer solutions were presented and (2) more precision for each solution became available at each ensuing display. Therefore, the TO Display had the greatest capability to compare or trade off solutions, and the RTED Display had the least capability. The TO Display provided the user with the capability to determine and compare the behavior of solutions over a wide range of maneuver times. Computation times were reasonable because the solutions were not integrated. However, the solutions were sufficiently accurate for selecting the possible candidates for the real-time situation. Then, the prime-candidate solutions selected by the flight controller were recomputed with more precision in the AST. From the candidate solutions, the one solution that was most appropriate was generated in the RTED Display and became available for incorporation into the planned trajectory.

Early Program Development

Apollo Trajectory Decision Logic Prototype Program. - During 1964 and 1965, a prototype real-time-trajectory planning and evaluation program, known as the Apollo Trajectory Decision Logic Prototype (ATDLP) Program, was developed. Various modules of the ATDLP Program comprised the abort-maneuver logic for a lunar-landing-type mission. The definition of this abort-program logic occurred at a series of meetings between MSC and contractor personnel. The specification of this logic originated at these meetings as well as from existing MSC and prime-contractor program logic and analysis.

The ABORT Program. - Concurrently with the ATDLP Program, a program known as ABORT was being developed to serve as a real-time-program candidate and as a general abort-maneuver-analysis tool. The solution modes for the abort-processor part of the ATDLP Program generally paralleled those of the ABORT Program. The ABORT Program contained both earth- and moon-centered conic-solution logic as well as precision-integrated-coast logic and finite-burn-solution logic. The solution characteristics of each mode in the ABORT Program were determined from conic propagations by scanning over the key trajectory parameters. Therefore, optimum solutions were conic optimums and not necessarily precision optimums.

The MINMAX Program. - A third program, called MINMAX, which had the capability of solving abort problems and contributed logic to the ATDLP Program, was under development. The MINMAX Program solved trajectory-planning problems by an optimization scheme rather than by a scan approach. To obtain a solution, one set of variables (independent variables) was adjusted so that a second set of variables (dependent variables) would acquire a desired set of values. The MINMAX Program also had the capability of generating precision-integrated coasting arcs as well as conic or patched conic arcs. For the MINMAX Program, the solution mode or characteristics were determined by the choice of dependent variables. A characteristic of this program was

that the attainment of a solution depended heavily upon good initial values for the independent variables and a proper set of weights for the choice of variables.

Comparison of the ABORT and MINMAX Programs as real-time candidates. - An evaluation of the ABORT and MINMAX Programs was initiated in the fall of 1965. The objective of the evaluation was to determine the capabilities and deficiencies of each program as candidate real-time abort-solution logic. The following types of solutions in both earth and moon reference were generated by the ABORT Program for the evaluation.

1. Time- and fuel-critical unspecified area
2. Time- and fuel-critical specified site (PTP)
3. Time- and fuel-critical water landing

During the evaluation, logic was added to compute maneuvers that would return to an inaccessible PTP with the smallest miss distance. Before this addition, only PTP solutions that always hit the target site with zero miss distance were generated. Also, all three solution modes were generated for some cases by using the precision logic of the ABORT Program as well as the MINMAX Program.

From late 1965 to early 1966, the results and solutions that were obtained by this evaluation were being discussed by MSC and contractor personnel. Objectives were the understanding of the behavior of the solutions, the development of a method by which the solution behavior could be displayed, and the determination of which solution parameters should be presented to the flight controller.

Initial Formulation of Abort Procedures

Preliminary operational program requirements. - In 1965, a second group of MSC and contractor personnel was assigned to formulate abort procedures and to design the supporting real-time displays. Early in 1966, the initial results were presented to MSC representatives concerned with flight control and flight dynamics. The abort procedures implied a highly automatic RTCC processor and would necessarily have been included in the logic of the program to acquire the desired automation. The resulting logic would produce different types of solution, depending upon the abort situation. At the same time, the RTE Program would contain a look-ahead feature in which the flight controller would anticipate abort situations by generating possible future solutions along the planned trajectory. Two types of abort displays were presented: (1) a graphical TO Display, called the Advanced Planning TO, which presented maneuver ΔV as a function of total trip time and maneuver time, and (2) a tabular display, called the Abort Mode Selection Display, which presented a selected set of parameters for the various solutions that would be available for a particular maneuver time.

Two general guidelines for the procedures group and the program-development group emerged early in 1966. First, the processor would not be highly automatic, although selection of the desired type of solution by the flight controller would be possible. Second, the abort procedures would not constrain the processor, but the real-time program would have a general solution capability.

Modifications of TO Display requirements. - As the analysis by the program-development group continued, the need for some type of TO Display became apparent. This graphical or analog display was better suited than a digital display to presentation of the key parameters for a large number of solutions. In addition to the maneuver ΔV and the landing time, the miss distance for the PTP solutions was observed to be a key parameter that was needed for solution comparison. Therefore, the original TO Display was modified so that maneuver ΔV , landing time, and miss distance would be presented for PTP mode solutions.

Basic Program Definition

Based on the suggestions of the procedures and flight control personnel, the program-development group defined the abort processor in 1966. The moon-centered logic of the ABORT Program would be used as the basis for the development of the moon-referenced conic logic. A new earth-centered conic abort logic was defined because of the desire to produce PTP solutions that could miss the landing site. These programs would generate solutions by scanning the key solution parameters rather than by implementing any simple iteration schemes for each mode. This decision resulted from a desire to generate solutions that were nonoptimum in the sense that they would be between TCUA and FCUA solutions. The two conic programs would provide the initial values of the maneuver (independent) variables for the MINMAX Program, and the selected parameters from the conic solution would provide the dependent variables. The MINMAX iterator was chosen because continued development of the iterator was assured by the availability of development personnel and because some success was attained by the iterator during the program evaluation.

Additional capabilities were added to the conic programs as a result of program evaluation and procedural discussions. The search-on-maneuver-time option was added to the moon-referenced, specified-site options. The non-zero-miss-distance solution was implemented into the PTP mode for earth- and moon-referenced programs. The minimum-miss-distance PTP mode was developed in addition to the existing modes for the earth-referenced program. The ATP mode was added to earth- and moon-referenced programs; the water-landing mode was dropped. For the remainder of 1966 and throughout 1967, the development of both programs continued. The accuracy of solutions was enhanced by the addition of ΔV calibration logic to both programs, and the reliability and computation speed of the programs were increased.

The RTE Program logic was defined formally in the fall of 1966; the earth-centered conic logic was defined in October 1966; and the moon-centered conic logic was defined in February 1967. At this time, no programming requirements existed for any of the precision-trajectory computational logics or for the overall supervisory logic. Basically, the defined logic was the first-guess logic for the precision part of the program. This logic defined solution modes in three categories.

1. In earth and moon reference: a TCUA and an FCUA, an ATP and PTP trade-off, a time- and fuel-critical PTP subject to a maximum miss distance, and a time- and fuel-critical ATP
2. In earth reference: an EFCUA and a minimum-miss-distance PTP
3. In moon reference: a search option for the ATP and PTP modes

Approximately 1 year after the initial definition of the program logic, several factors precipitated a redefinition of the logic. During the year, logic deficiencies were discovered and corrected, and some program limitations were removed. The program-development group continued to discuss the operation and application of the processor modes; the relative importance of the processor modes and the amount of redundancy among the modes were evaluated. As a result, the moon-referenced TCUA, the earth-referenced minimum-miss-distance PTP, and the time-critical PTP modes were deleted from the processor in the second program definition.

When the MINMAX logic was chosen as the basis for generating the precision solutions from the conic solution, no specific effort was initiated toward the selection of independent and dependent variables for the various conic-solution modes. General investigations were continued into the behavior of the abort solutions over a wide range of conditions and into the behavior of the various finite-burn guidance laws. In particular, reliability and primary control parameters of the guidance laws were investigated. The Lambert cross-product-steering law required extensive investigation to determine a target vector placement scheme sufficiently general in scope to support a nonnominal abort maneuver. These investigations continued through the spring of 1968.

Beginning in 1967, studies of the MINMAX-iterator variable setup began in earnest. The problem was to select those key parameters from the conic solution that, when used as dependent variables, would reproduce the desired characteristics of the conic solution with an acceptable ΔV penalty. Reliably generated solutions that converged quickly to the final solution values were needed. An initial definition of the precision-computation logic and the interfacing logic between the conic and precision logic were attempted in April 1967. The precision-computation logic to support the AST and RTED and the supervisory logic of the abort processor were defined in December 1967 and February 1968, respectively. At this time, only two guidance laws, the external ΔV and Lambert's targeting in conjunction with cross-product steering, were defined for the RTED Display logic.

During late 1967, emphasis was shifted from the PTP mode to the ATP mode because of discussions among personnel, results from dispersion analysis, and shorter entry ranges being planned for lunar-return energies. The TEMC target objectives shifted from specified site to FCUA. The AST FCUA mode, although optimum in the conic mode, was not truly optimum in the precision mode and, thus, was not able to support TEMC targeting. The earth-referenced FCUA mode was altered to allow ΔV minimization in the precision (AST) mode by using the MINMAX iterator. Because schedules did not allow the development of any other precision FCUA mode, the increased optimality was acquired at the cost of a computer-run-time penalty and a lower convergence reliability. The moon-referenced FCUA was not altered in this manner because this optimization mode would not reliably yield convergence for moon-referenced cases. Two manual techniques were developed to increase the optimality of the moon-referenced FCUA mode. In moon reference, the ATP mode would yield a more accurate FCUA solution by iterating on the longitude of the FCUA solution to lower the maneuver ΔV . In addition, an investigation into the direction of the precision FCUA maneuver ΔV revealed that, for TEMC maneuvers, the ΔV was applied along the local horizontal relative to the earth. This maneuver attitude was also generally true for earth-referenced, corridor-control TEMC. To allow the applications of this knowledge in the RTCC, the manual-input option of the RTED was modified to allow the maneuver to be specified in an earth-referenced local-vertical/local-horizontal coordinate system.

The RTE Program in the Apollo 8 Mission

The establishment of the testing and program delivery schedules for the Apollo 8 mission in December 1968 caused the delay of checkout and the implementation of some options in the RTE Program. Neither the PTP mode nor Lambert's guidance would be verified for the Apollo 8 mission. Also, the moon-referenced search options would not be incorporated into the logic. The Apollo 8 RTE Program consisted of the three RTE displays and the following AST discrete solution modes: earth-centered TCUA, fuel-critical ATP, and FCUA.

The RTE Program after the Apollo 8 Mission

For the Apollo 10 mission in May 1969, the moon-referenced search options and the PTP mode were completed and verified. Lambert's guidance was never completed and, eventually, was eliminated from the program. Also, a new set of independent and dependent variables for the precision iterator setups was defined to improve speed, accuracy, and reliability. Between the Apollo 10 and 11 missions, no major change was made in the abort processor.

Observations

Three general observations seem to be apparent in retrospect. First, considerable initial effort was expended on the moon- and earth-referenced conic logics to develop several modes. Ultimately, approximately one-third of these modes were discarded during final program development. Because the decision that the conic logic would supply initial values to the MINMAX iterator was made at the beginning of program development, the development seemed to be excessively dependent on this mode. Second, although the accuracy and optimization in the conic logic were always improved, the means of optimization in the precision logic was not considered. Only recently has the program expanded with the development of the gradient-check method for earth-referenced FCUA solutions and the application of the state vector offset technique that had been developed for the translunar midcourse processor. Third, because of the late start on the development of the precision computation logic for the RTE Program, time was available for the development of only the agreed-upon approach.

ONBOARD SOFTWARE CONSIDERATIONS

Onboard Midcourse Targeting

The onboard midcourse routines were developed mainly at MIT, but the routines were given considerable evaluation at MSC. Schemes for TLMC and TEMC essentially attempted to place the spacecraft on a nominal trajectory. Virtually no interdevelopment existed between the onboard targeting techniques and the RTCC targeting techniques, although all personnel were aware of the developments within each area.

In the mid-1960's, the two onboard midcourse targeting routines were among the first to be eliminated from the onboard computer because of increasing demands for



onboard computer storage and because of the decision that nearly all targeting for maneuvers during the lunar mission would be conducted from the ground control center. Therefore, an onboard TLMC program was unnecessary because the program would be used only if a malfunction of the ground equipment or a loss of communications occurred, in which case it was assumed that an abort maneuver would be required or a midcourse maneuver would be delayed until such time as the ground facility had the capability to solve the problem. Essentially the same reasoning applied to the TEMC targeting routine, except that a targeting program would be needed if communications were lost. Such a program already existed in the form of the onboard RTE Program. Therefore, the TEMC Targeting Program was eliminated from consideration for the spacecraft computer.

Onboard Return-to-Earth Targeting

The onboard RTE Program was developed mainly at MIT. At the outset of the Apollo Program, MIT and MSC personnel agreed on the program requirements and began development. The same MSC personnel were also responsible for the development of the RTCC RTE Program, so that some interdevelopment between the onboard and ground targeting schemes existed. Eventually, personnel at MSC were forced to devote practically all of their efforts to the development of the RTCC RTE Program. The MSC personnel, however, continued to monitor development of the onboard program and maintained an in-house simulation corresponding to the MIT version of the program. An extensive effort was made by MSC personnel to verify the onboard program. As a part of this verification, the onboard results were compared with RTCC results for similar cases. When the two programs had similar constraints and the same inputs, the results of the two programs agreed. The RTCC RTE Program necessarily had the capability of providing a greater variety in types of solution. The onboard system had a fuel-critical and a time-critical mode. Inputs could be varied in the time-critical mode to change landing longitude.

The onboard RTE Program was also affected by the increased demands upon onboard computer storage space. Originally, moon- and earth-reference logic had been in the program. Because the Mission Control Center could supply block-data-type solutions to the spacecraft before the vehicle entered the lunar sphere, the moon-centered part of the program was one of the first items deleted from the computer.

The only serious deficiency in the program was uncovered during verification of the onboard RTE Program for the Apollo 8 mission. An implicit constraint on the entry speed existed in the program. This limit was originally thought to be the parabolic speed at the altitude of entry interface; however, shortly before launch, the verification effort showed that the limit was considerably lower than the parabolic value. This low limit meant that, for several cases, many of which were nominal return trajectories from the moon, the onboard system would call for a maneuver merely to increase the RTE time, even when the spacecraft was on a satisfactory trajectory. A workaround procedure was developed by personnel of MSC, MIT, a contractor, and the flight crew. A crew checklist was devised for the procedure, and the procedure was verified a few days before lift-off. This limit in entry speed was corrected to a sufficiently large value for the Apollo 10 onboard computer program. The verification efforts at MSC have shown no serious errors in the program since that time.

CONCLUDING REMARKS

The development of targeting techniques for the translunar injection, translunar midcourse, lunar-orbit insertion, and return-to-earth maneuvers in the Apollo lunar missions has been documented. The result of the targeting-technique development was a set of computer programs that is used in the real-time computer complex to calculate the guidance parameters for each maneuver. All of these programs, except the translunar-injection targeting-update processor, have been used successfully to target maneuvers on each Apollo mission thus far.

The development of each of the targeting programs was heavily dependent upon other developments that took place concurrently in the Apollo Program (e. g., guidance-equation development, mission-profile evolution, flight-hardware specifications, etc.). Consequently, the requirements for the targeting program changed as guidance equations were changed, as the mission profile changed, and so forth. In one instance, this proved to be beneficial: the existing lunar-orbit-insertion targeting processor has evolved from the final definitions of the mission profile (the two-burn lunar-orbit insertion) and of the onboard guidance equations (which call for constant-attitude maneuvers).

The other three targeting programs depended heavily upon the Generalized Forward Iterator, and, as the name implies, generalized iterative techniques were used extensively. Nevertheless, differences existed among the three programs. Development of the Return-to-Earth Targeting Program was based on the intention to use, first, several conic programs and, finally, the iterator to search for the final solution. The translunar midcourse processor used several steps to arrive at a solution, and each step contained some iteration. The same process existed for the translunar-injection targeting-update processor.

Considerable knowledge has been gained in the development of these targeting techniques and computer programs. Each system has been constantly refined, and new analytical methods are still being evolved.

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